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Martin

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[54] **BOWED TURBINE BLADE**

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[51] **Int. Cl.**[4] **F01D 5/12**

[52] **U.S. Cl.** .... **416/223 A;** 416/242; 416/DIG. 2; 415/181

[58] **Field of Search** .... 416/223 A, DIG. 2, 242; 415/181

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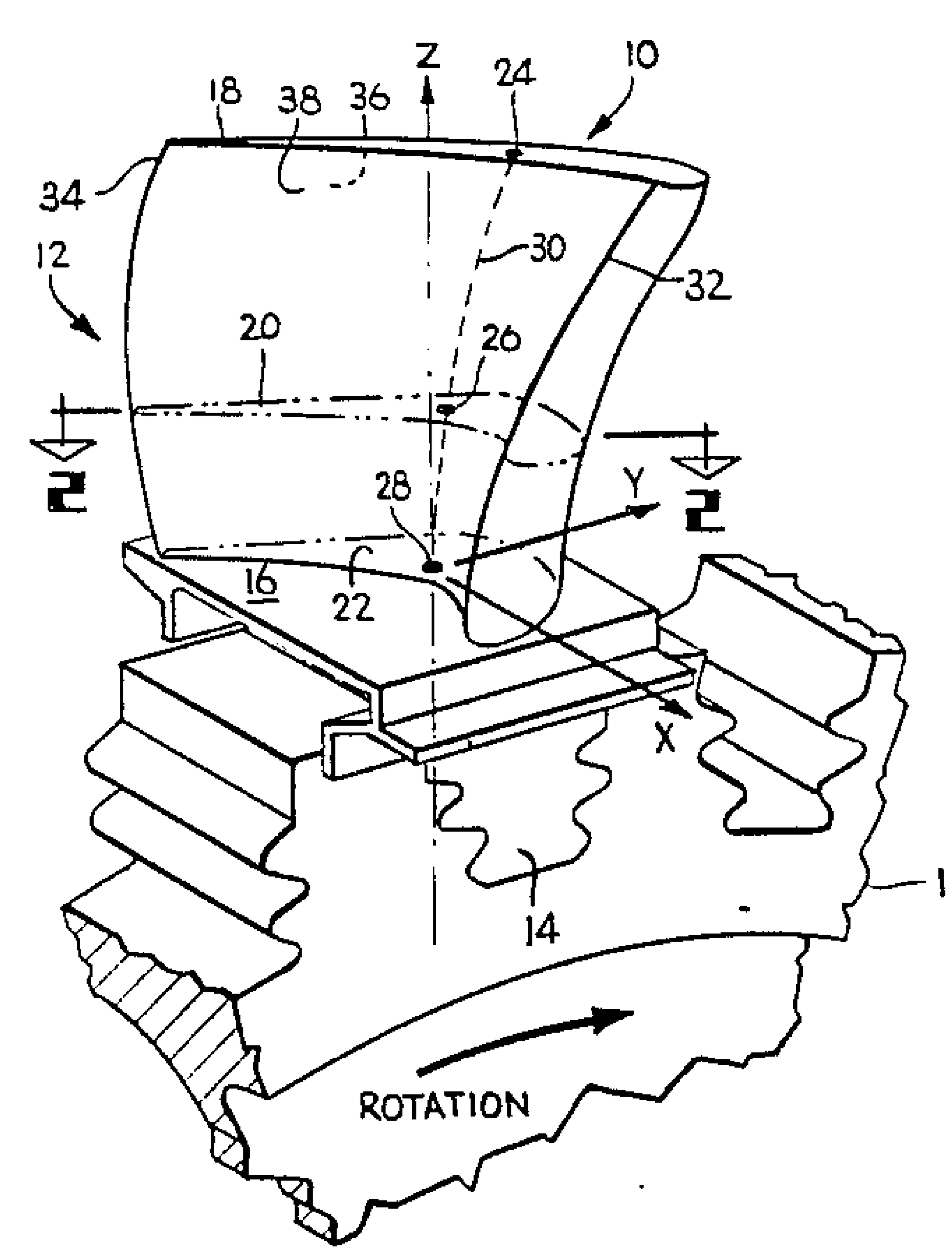
[57] **ABSTRACT**

The invention comprises a blade for a gas turbine engine including an airfoil portion having a non-linear stacking axis which is effective for generating a compressive component of bending stress due to centrifugal force acting on the blade. The compressive component of bending stress is provided in a life-limiting section of the blade, which, for example, includes trailing and leading edges of the blade. Inasmuch as the stacking axis, which represents the locus of centers of gravity of transverse sections of an airfoil portion of the blade, is non-linear, an increased amount of a compressive component of bending stress can be generated at a life-limiting section between root and tip sections of the blade without substantially increasing bending stress at the root of the blade.

9 Claims, 4 Drawing Figures



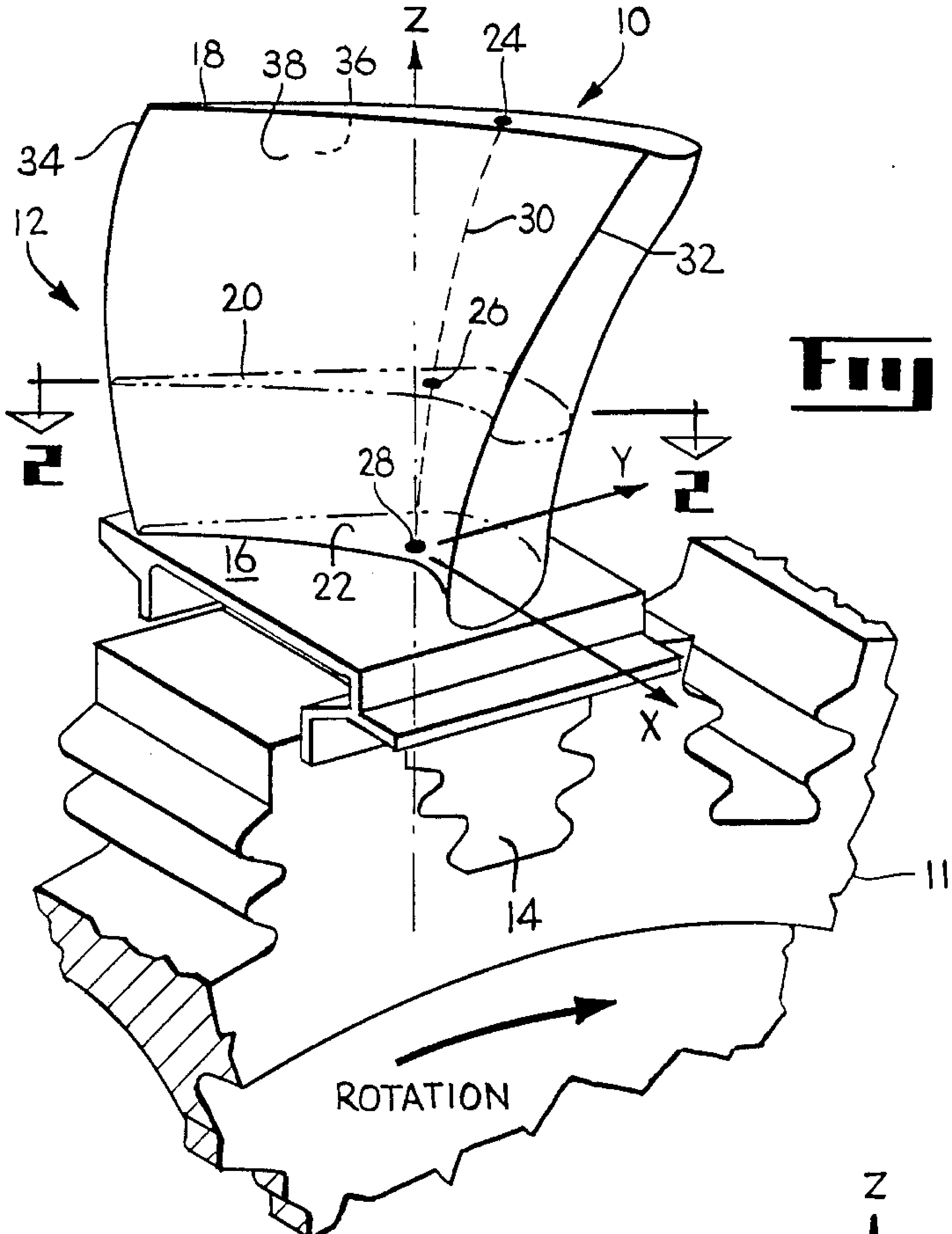


Fig 1

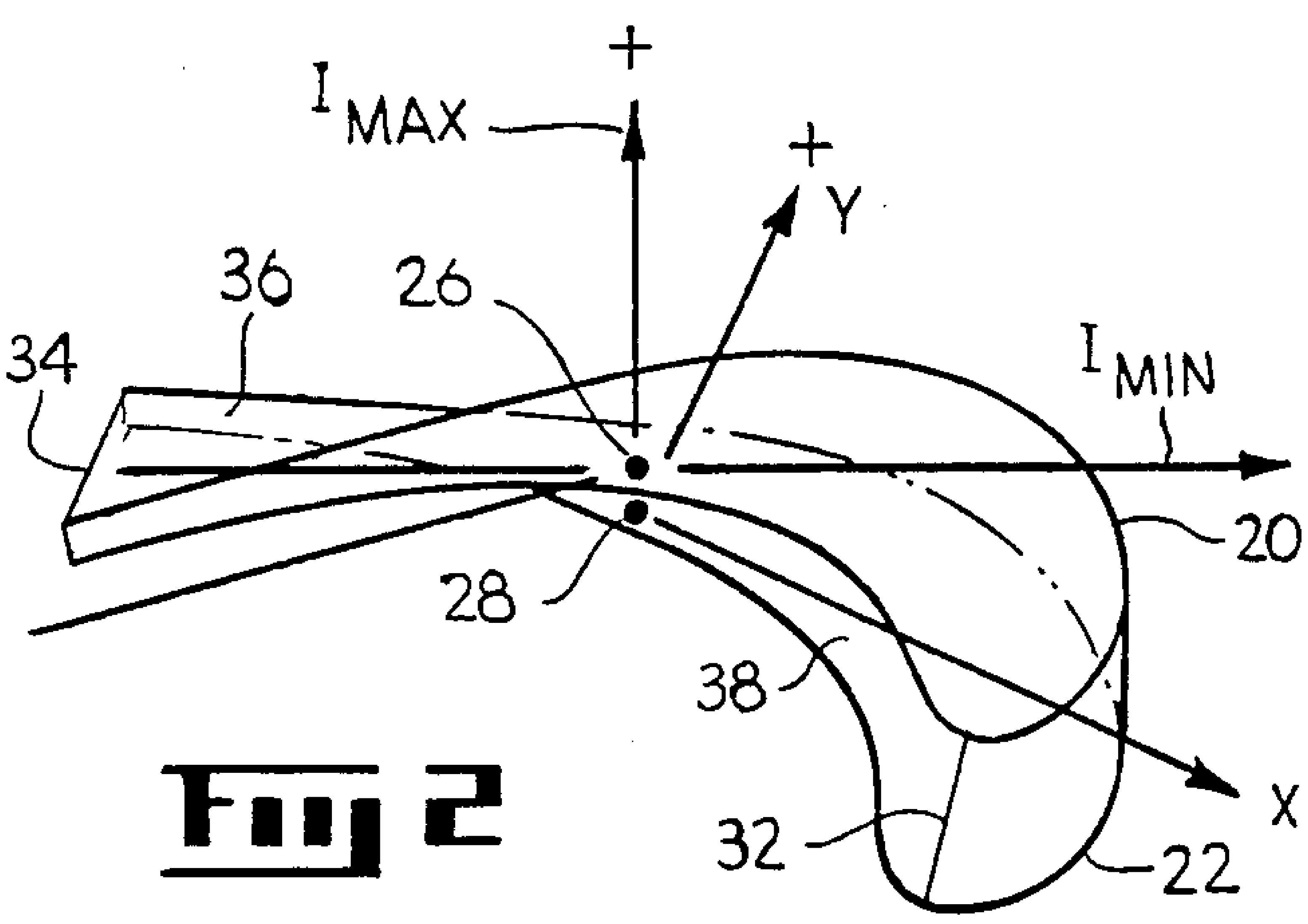


Fig 2

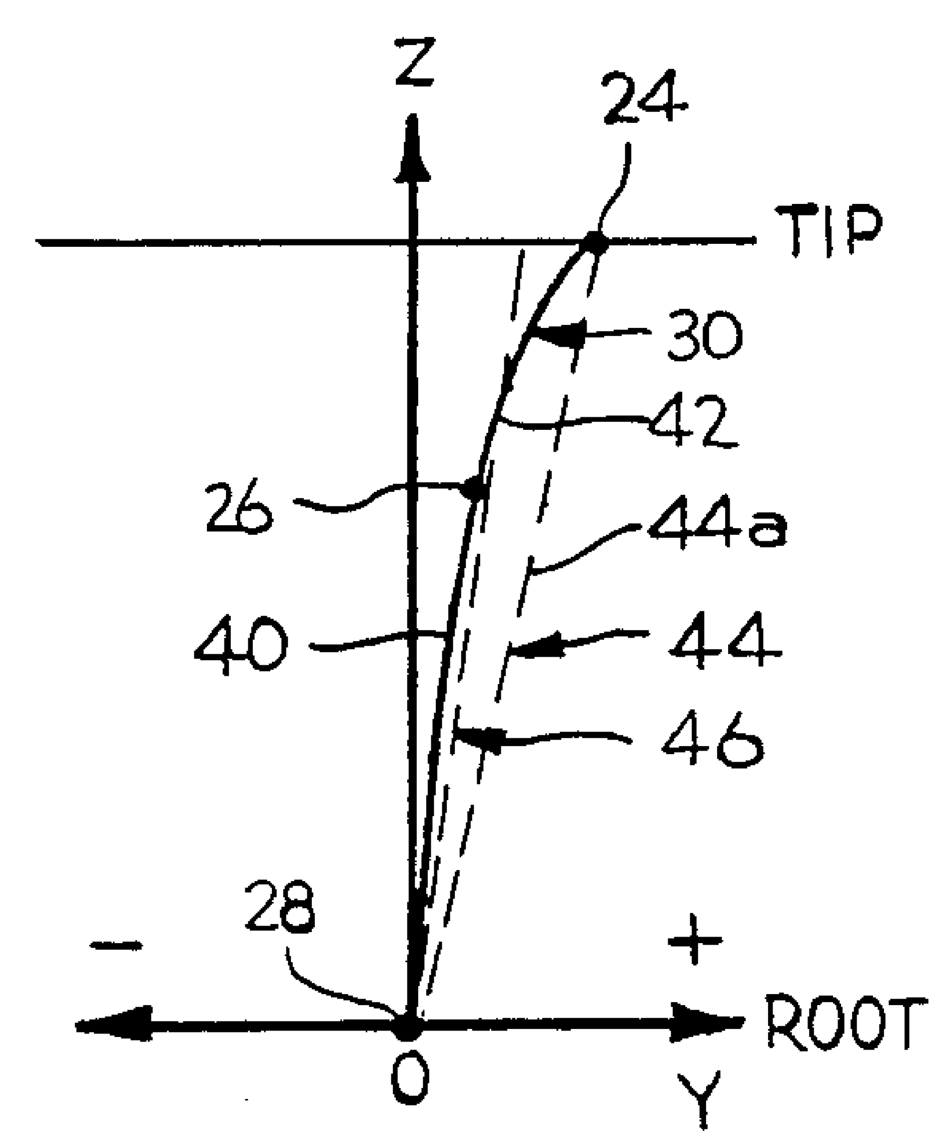


Fig 3

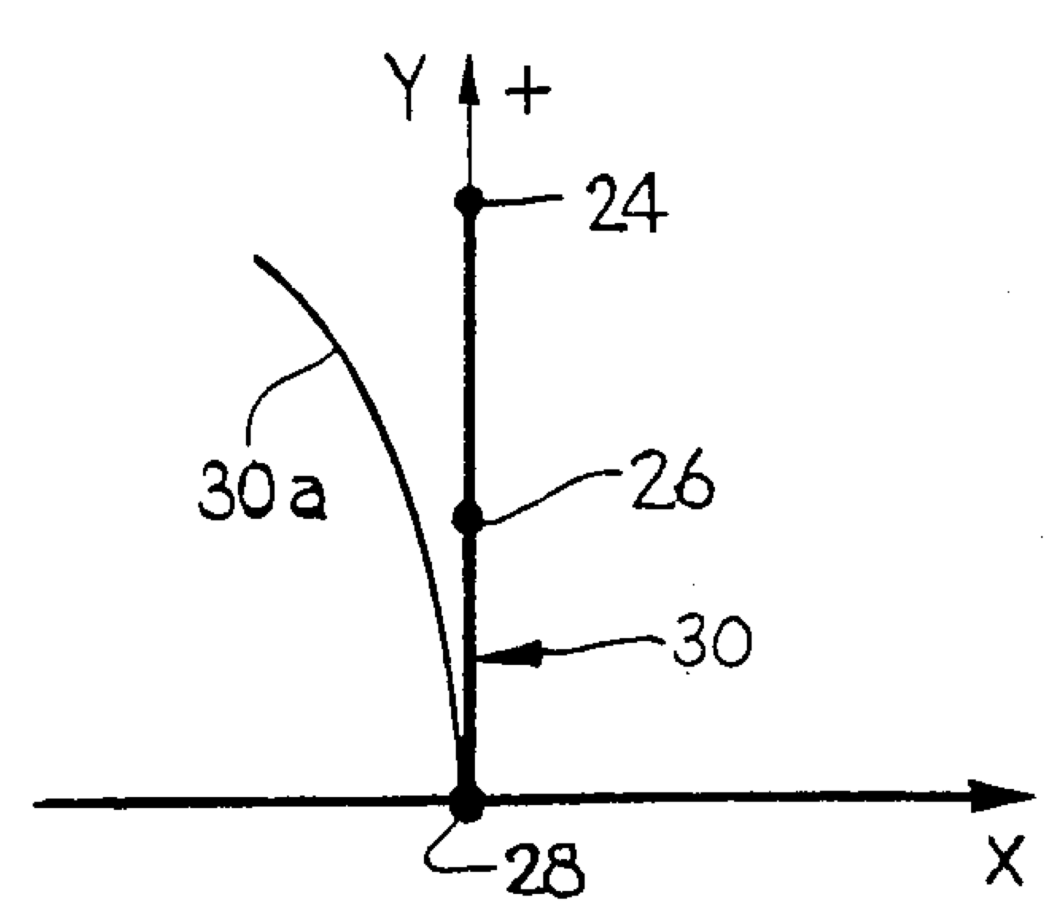


Fig 4

BOWED TURBINE BLADE

The Government has rights in this invention pursuant to Contract No. N00019-76-C-0261 awarded by the Navy.

CROSS REFERENCE TO A RELATED APPLICATION

This application is related to application Ser. No. 560,656, now U.S. Pat. No. 4,585,395, issued Apr. 29, 1986, and assigned to the same assignee as the instant application.

BACKGROUND OF THE INVENTION

This invention relates generally to blades for a gas turbine engine and, more particularly, to an improved blade effective for reducing stresses due to centrifugal force to improve the useful life of the blade.

An axial flow gas turbine engine conventionally includes a plurality of rows of alternating stationary vanes and rotating blades. The rotating blades are typically found in fan, compressor, and turbine sections of the engine, and inasmuch as these blades rotate for performing work in the engine, they are subject to stress due to centrifugal forces.

The centrifugal stress in a blade is relatively substantial and includes a substantially uniform centrifugal tensile stress and centrifugal bending stress including a tensile component and a compressive component which are added to the uniform tensile stress.

In a turbine section of the gas turbine engine, turbine blades are also subject to relatively hot, pressurized combustion gases. These gases induce bending stresses due to the pressure of the combustion gases acting across the turbine blades, which stresses are relatively small when compared to the centrifugal stresses. The relatively hot gases also induce thermal stress due to any temperature gradient created in the turbine blade.

A turbine blade, in particular, has a useful life, i.e., total time in service after which time it is removed from service, conventionally determined based on the above-described stresses and high-cycle fatigue, low-cycle fatigue, and creep-rupture considerations. A typical turbine blade has an analytically determined life-limiting section wherein failure of the blade is most likely to occur. However, blades are typically designed to have a useful life that is well in advance of the statistically determined time of failure for providing a safety margin.

A significant factor in determining the useful life of a turbine blade is the conventionally known creep-rupture strength, which is primarily proportional to material properties, tensile stress, temperature, and time. Notwithstanding that the relatively high temperatures of the combustion gases can induce thermal stress due to gradients thereof, these temperatures when acting on a blade under centrifugal tensile stress are a significant factor in the creep consideration of the useful life. In an effort to improve the useful life of turbine blades, these blades typically include internal cooling for reducing the temperatures experienced by the blade. However, the internal cooling is primarily most effective in cooling center portions of the blade while allowing leading and trailing edges of the blade to remain at relatively high temperatures with respect to the center portions thereof. Unfortunately, the leading and trailing edges of the blade are also, typically, portions of the blade subject to the highest stresses and therefore, the life-limiting section of a blade typically occurs at either the leading or trailing edges thereof.

Furthermore, a primary factor in designing turbine blades is the aerodynamic surface contour of the blade which is typically determined independently of the mechanical strength and useful life of the blade. The aerodynamic performance of a blade is a primary factor in obtaining acceptable performance of the gas turbine engine. Accordingly, the aerodynamic surface contour that defines a turbine blade may be a significant limitation in the design of the blade from a mechanical strength and useful life consideration. With this aerodynamic performance restriction, the useful life of a blade may not be an optimum, which, therefore, results in the undesirable replacement of blades at less than optimal intervals.

Accordingly, it is an object of the present invention to provide a new and improved blade for a gas turbine engine.

Another object of the present invention is to provide an improved turbine blade effective for reducing tensile stress in a life-limiting section of the blade by adding a compressive component of bending stress thereto.

Another object of the present invention is to provide an improved turbine blade having improved useful life without substantially altering the aerodynamic surface contour of the blade.

Another object of the present invention is to provide an improved turbine blade wherein tensile stress is reduced in a life-limiting section thereof without substantially increasing stress in other sections of the blade.

SUMMARY OF THE INVENTION

The invention comprises a blade for a gas turbine engine including an airfoil portion having a non-linear stacking axis which is effective for generating a compressive component of bending stress due to centrifugal force acting on the blade. The compressive component of bending stress is provided in a life-limiting section of the blade, which, for example, includes trailing and leading edges of the blade. Inasmuch as the stacking axis, which represents the locus of centers of gravity of transverse sections of an airfoil portion of the blade, is non-linear, an increased amount of a compressive component of bending stress can be generated at a life-limiting section between root and tip sections of the blade without substantially increasing bending stress at the root of the blade.

BRIEF DESCRIPTION OF THE DRAWINGS

The invention, together with further objects and advantages thereof, is more particularly described in the following detailed description taken in conjunction with the accompanying drawings in which:

FIG. **1** is a perspective view of an axial entry blade for a gas turbine engine.

FIG. **2** is a sectional view of the blade of FIG. **1** taken along line **2**—**2**.

FIG. **3** is a graphical representation of the stacking axis of the blade of FIG. **1** in a Y-Z plane.

FIG. **4** is a graphical representation of the stacking axis of the blade of FIG. **1** in an X-Y plane.

DETAILED DESCRIPTION

Illustrated in FIG. **1** is a generally perspective view of an exemplary axial-entry turbine blade **10** mounted in a turbine disk **11** of a gas turbine engine (not shown).

The blade **10** includes an airfoil portion **12**, a dovetail portion **14** and an optional platform **16**. The airfoil portion **12** of the blade **10** comprises a plurality of transverse sections including a tip section **18**, an intermediate section **20** and a root section **22**, each of which has a center of gravity (C.g.) **24**, **26** and **28**, respectively. The locus of the centers of gravity of the airfoil portion **12** define a stacking axis **30**, which in accordance with the present invention is non-linear, e.g. bowed, and is described in further detail below.

The blade **10** further includes a conventional reference XYZ coordinate system having an origin at the C.g. **28** of the root section **22**. This coordinate system includes: an X, axial axis, which is aligned substantially parallel to a longitudinal centerline axis of the gas turbine engine; a Y, tangential axis, which is normal to the X axis and has a positive sense in the direction of rotation of the turbine disk **11**; and a Z, radial axis, which represents a longitudinal axis of the blade **10** which is aligned coaxially with a radial axis of the gas turbine engine.

As illustrated in FIGS. **1** and **2**, the airfoil portion **12** of the blade **10** has an aerodynamic surface contour defined by and including a leading edge **32** and a trailing edge **34**, between which extend a generally convex suction side **36** and a generally concave pressure side **38**. The pressure side **38** faces generally in a negative direction with respect to the reference tangential axis Y; the suction side **36** faces generally in a positive direction with respect thereto.

Each of the plurality of transverse sections of the airfoil portion **12** of the blade **10** has its own conventionally known principal coordinate system. Illustrated in FIG. **2** is an exemplary principal coordinate system for the intermediate section **20** including an I_{max} axis and an I_{min} axis. The principal coordinate system has an origin at the C.g. **26** of the intermediate section **20**. I_{max} represents an axis of maximum moment of inertia about which the intermediate section **20** has a maximum stiffness or resistance to bending and I_{min} represents an axis of minimum moment of inertia about which the intermediate section **20** has a minimum stiffness or resistance to bending.

A conventional method of designing the blade **10** includes designing the airfoil portion **12** for obtaining a preferred aerodynamic surface contour as represented by the suction side **36** and the pressure side **38**. The stacking axis **30** of the airfoil portion **12** would be conventionally made linear and coaxial with the reference radial axis Z. A suitable dovetail **14** and an optional platform **16** would be added and the entire blade **10** would then be analyzed for defining a life-limiting section, which, for example, may be the intermediate section **20**, which is typically located between about 40 percent to about 70 percent of the distance from the root **22** to the tip **18** of the airfoil portion **12**. Of course, analyzing the blade **10** for defining a life-limiting section is relatively complex and may include centrifugal, gas and thermal loading of the blade **10**, which is accomplished by conventional methods.

However, in accordance with the present invention, the method of designing the blade **10** further includes redesigning the blade having the linear stacking axis, i.e., the reference blade, for obtaining a non-linear, tilted stacking axis **30** which is effective for introducing a compressive component of bending stress in the predetermined, life-limiting section.

More specifically, it will be appreciated from an examination of FIGS. **1** and **2** that if the stacking axis **30** is spaced from the reference radial axis Z, that upon centrifugal loading of the airfoil portion **12**, centrifugal force acting on the centers of gravity, C.g. **26** for example, will tend to rotate or bend the stacking axis **30** toward the reference radial axis Z thus introducing or inducing bending stress.

It will be appreciated from the teachings of this invention, that by properly tilting and spacing the stacking axis **30** with respect to the reference radial axis Z a compressive component of bending stress can be induced at both the leading edge **32** and the trailing edge **34** of the intermediate section **20** due to bending about the I_{min} axis as illustrated in FIG. **2**. Of course, due to equilibrium of forces, an off-setting tensile component of bending stress is simultaneously introduced in the suction side **36** of the intermediate section **20** and generally at positive values of the I_{max} axis.

Illustrated in more particularity in FIG. **3** is an exemplary embodiment of the stacking axis **30** in accordance with the present invention and as viewed in the Y-Z plane. The stacking axis **30** extends away from and is spaced from the reference radial axis Z in a positive direction with respect to the reference tangential axis Y from, but not including, the root section **22** to the tip section **18**. The stacking axis **30** is generally defined as including a first portion **40** extending from the C.g. **28** of the root section **22** to the C.g. **26** of the intermediate section **20**, and a second portion **42** extending from C.g. **26** of the intermediate section **20** to the C.g. **24** of the tip section **18**. Also illustrated is a reference, linear, tilted stacking axis **44** extending from C.g. **28** of the root section **22** to the C.g. **24** of the tip section **18**. The stacking axis **30** has an average slope represented by dashed line **46** which, as illustrated, is larger in magnitude than the slope of the reference axis **44** and is disposed between the reference radial axis Z and the reference stacking axis **44**.

Assuming, for example, that the life-limiting section of the airfoil portion **12** is located at the intermediate section **20** it will be apparent from the teachings herein that a compressive component of bending stress can be introduced in the intermediate section **20** by using either the linear stacking axis **44** or the non-linear stacking axis **30**. To introduce the desired bending stress at the intermediate section **20**, the stacking axis **30** must be tilted with respect to the reference radial axis Z at those sections radially outwardly from the intermediate section **20**, i.e., the second portion **42** of the stacking axis **30**. The slope of the stacking axis **30** is generally inversely proportional to the amount of bending stress realizable at the intermediate section **20**.

As illustrated in FIG. **3**, the first portion **40** has a first, average slope, and the second portion **42** has a second, average slope, the first slope being greater than the second slope. This is effective for obtaining increased bending stress at the intermediate section **20** without adversely increasing bending stress at the root section **22**. Additionally, the second portion **42** of the stacking axis **30** has less of a slope than a comparable portion **44*a*** of the reference linear stacking axis **44**, which indicates that more bending stress can be introduced thereby at the intermediate section **20**.

However, not only is the reference linear stacking axis **44** less effective in introducing the desired bending stress to the intermediate section **20**, but inasmuch as the reference stacking axis **44** is linear from C.g. **28** to the

C.g. **24**, substantial, undesirable bending stresses are also introduced at the root section **22**. These increased bending stresses at the root **22** are a limit to the amount of bending stress introducible by the reference linear stacking axis **44** in the life-limiting section of the airfoil portion **12** in that the life-limiting section may thereby be relocated from the intermediate section **20** to the root section **22**.

In contrast, inasmuch as the average slope line **46** for the non-linear stacking axis **30** has a greater magnitude than that of the reference stacking axis **44**, it will be appreciated that not only does the non-linear stacking axis **30** provide for increased bending stress at the intermediate section **20** but less of a bending stress at the root **22** as compared to that provided by the reference linear stacking axis **44**. Accordingly, a non-linear stacking axis **30** is more effective for introducing the desired compressive components of bending stress at the life-limiting section without adversely increasing the bending stresses at the root section **22**.

FIG. **3** illustrates in more particularity the non-linear stacking axis **30** according to the present invention. The stacking axis **30** is described as being non-linear from the C.g. **28** of the root section **22** to the C.g. **24** of the tip section **18** and may include either linear or curvilinear portions therebetween.

As long as the stacking axis **30** has portions which extend away from and are spaced from the reference radial axis Z in a positive direction with respect to the reference tangential axis Y compressive components of bending stress will be introduced at the leading edge **32** and the trailing edge **34** of the airfoil portion **12**.

Optimally, the magnitude of compressive stress induced is preferably made equal to approximately the compressive yield strength of the blade material. This will provide maximum compressive stress in the leading edge **32** and the trailing edge **34** during operation which will provide improved fatigue life. Furthermore, the stacking axis **30** can be tilted also to induce stresses initially greater than the compressive yield strength, which stresses will yield thereto after the first few initial cycles of operation, so that manufacturing inaccuracies do not prevent the induced stress from reaching the optimal value.

More specifically, and as additionally illustrated in FIG. **2**, the tangential reference axis Y is generally aligned with the I_{max} axes of the transverse sections of the airfoil portion **12**, the I_{max} axis of the intermediate section **20**, for example. Accordingly, in operation, centrifugal forces act at each of the centers of gravity of the airfoil portion **12** and will thus tend to straighten the airfoil portion **12** to bring the stacking axis **30** closer to the reference radial axis Z. For example, when the average slope line **46** of the stacking axis **30** is spaced from reference radial axis Z in a generally positive direction with respect to the tangential axis Y and the I_{max} axis, compressive components of bending stress will be introduced at the leading edge **32** and trailing **34**.

Illustrated in FIG. **4** is a view of the stacking axis **30** with respect to the X-Y plane. The stacking axis **30** preferably lies substantially in a plane defined by the reference radial axis Z and tangential axis Y, is preferably linear in the X-Y plane and is preferably aligned along the positive Y axis. This is preferred so that the aerodynamic surface contour and orientation of the airfoil portion **12** does not significantly change as the stacking axis **30** is tilted.

Alternatively, the spacing of the stacking axis **30** from the reference radial axis Z could also be positive in magnitude and be substantially oriented along the I_{max} direction for each of the transverse sections and might look like a stacking axis **30***a* illustrated in FIG. **4**. However, the relative twist of the airfoil portion **12**, i.e., its orientation with respect to the reference axial axis X, would change from that of an untilted blade, thusly changing the aerodynamic surface contour of the airfoil portion **12**.

While there have been described what are considered to be preferred embodiments of the present invention, other embodiments will be apparent from the teachings herein and are intended to be covered by the attached claims.

Having thus described the invention, what is desired to be secured by Letters Patent of the United States is:

1. A blade for a gas turbine engine comprising an airfoil portion including a leading edge, a trailing edge, an intermediate section, and a non-linear stacking axis effective for introducing a compressive component of bending stress, exceeding a compressive yield strength of said blade, in said trailing edge and said leading edge of said intermediate section due to centrifugal force acting on said blade.

2. A blade according to claim **1** wherein said airfoil portion further comprises:

a plurality of transverse sections including a root section, said intermediate section, and a tip section, each having a center of gravity;

reference radial and tangential axes extending outwardly from said center of gravity of said root section; and

wherein said stacking axis extends from said center of gravity of said root section and is spaced from said reference radial axis at said tip section.

3. A blade according to claim **2** wherein said stacking axis is spaced from said reference radial axis from said root section to said tip section.

4. A blade according to claim **2** wherein said airfoil portion further comprises:

a pressure side facing generally in a negative direction with respect to said reference tangential axis;

a suction side facing generally in a positive direction with respect to said reference tangential axis;

wherein said stacking axis extends away from said reference radial axis in a positive direction with respect to said reference tangential axis.

5. A blade according to claim **4** wherein said stacking axis lies substantially in a plane defined by said reference radial and tangential axes.

6. A blade according to claim **2** wherein said stacking axis further includes a first portion extending from said root section to said intermediate section and a second portion extending from said intermediate section to said tip section, said first portion having a first slope and said second portion having a second slope, said first slope being greater than said second slope.

7. A blade according to claim **2** wherein said airfoil portion further includes a predetermined life-limiting section, having an I_{min} axis and an I_{max} axis, and a suction side facing generally in a positive direction with respect to said I_{max} axis, and wherein said stacking axis is spaced from said reference radial axis in a positive direction with respect to said I_{max} axis.

8. A blade for a gas turbine engine comprising an airfoil portion including a leading edge, a trailing edge, a pressure side, and a suction side, and a plurality of transverse sections including a root section, an intermediate section, and a tip section, each of said plurality of sections having a center of gravity, the locus of which define a stacking axis, said blade further including reference radial and tangential axes extending outwardly from said center of gravity of said root section toward said tip section and said suction side, respectively, said stacking axis being non-linear and spaced from said reference radial axis from said intermediate section to said tip section and being effective for introducing a compressive component to bending stress in said trailing edge and said leading edge of said intermediate section due to centrifugal force acting on said blade, wherein said stacking axis includes a first portion extending from said root section to said intermediate section and a second portion extending from said intermediate section to said tip section, said first portion having a first slope and said second portion having a second slope, said first slope being greater than said second slope.

9. A blade for a gas turbine engine comprising an airfoil portion including a leading edge, a trailing edge, a pressure side, and a suction side, and a plurality of transverse sections including a root section, an intermediate section, and a tip section, each of said plurality of sections having a center of gravity, the locus of which define a stacking axis, said blade further including reference radial and tangential axes extending outwardly from said center of gravity of said root section toward said tip section and said suction side, respectively, said stacking axis being non-linear and spaced from said reference radial axis from said intermediate section to said tip section and being effective for introducing a compressive component of bending stress in said trailing edge and said leading edge of said intermediate section due to centrifugal force acting on said blade, wherein said stacking axis lies substantially only in a plane defined by said reference radial and tangential axes.

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